Magnetospheric Multi-Scale Mission
Requirements Specifications

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MMSqssREQS.doc

1 Introduction

The MMS mission consists of five identically spinning spacecraft acting as a single probe, flying in a tetrahedral configuration with spacing variable from less than 10km to several Re. Each spacecraft will contain an identical set of 3D instruments.

Onboard propulsion will allow the orbits of the MMS "probe" to have four separate mission phases, covering almost the entire magnetosphere, from near-Earth equator to the magnetotail. In each phase the 5-s/c "probe" will dwell at apogee in key boundary regions for~~of~~ magnetic reconnection and energy conversion.

2 Multiprobe Spacecraft

[MMS1] Five identically instrumented spacecraft shall comprise the MMS mission.

Table 1

2.1 Launch vehicle Compatibility

2.1.1 Vehicle

The mission design assumes that all 5 spacecraft will be launched from a single Delta 7925 with a 10' fairing.

2.1.2 Mass to orbit

The Delta 7925H-10 can launch 1540kg to the injection orbit. The PAF mass is not to be included in the 1540kg.

2.1.3 Launch date

Launch date is December 2004.

2.2 Definition of the orbit

Phase 0 (launch)
The launch vehicle inserts all 5 spacecraft into a 12Re x 1.2 Re 28° orbit. For the definitions of the orbits, the apogee & perigee distance from the center of the earth are given in units of earth radii. Inclinations are relative to the earth's equator.
Spacecraft are ejected spinning from the Delta third stage.

Time of launch set so that apogee is approximately at local midnight.

Phase I
The probes change orbit plane to 12Re x 1.2 Re <10° orbit
They stay in this orbit for ~8 months. The apogee yields long dwell times in the near tail region.
The probes initially form a tetrahedron with side 10km at apogee. During phase I the tetrahedron spacing is increased to 0.1Re.

Phase II
The probes change to 30Re x 1.2Re orbit in stages. The goal is to keep apogee in the dawnside of the magnetopause as the magnetic local time changes.
Probes form tetrahedron with side TBD.
Apogee remains in the nightside magnetosphere as the magnetic local time of the apogee changes from ~1000 to ~0400.
They stay in this orbit for ~4 months.

Phase III
Phase III Double L~~l~~unar ~~flyby changes~~Swingby (DLS) ~~orbit inclination to ~90~~<10° inclination. Outer loops will be in the magnetotail at approximately 127Re. The tetrahedron spacing will maximize during this phase at ~1-4Re near apogee to enable studies of plasmoidal structure. Two DLS maneuvers will be performed each lasting approximately one-month.

Phase IV changes to 50Re x 10Re polar orbit with apogee in the equatorial plane. In this orbit perigee skims the magnetopause from cusp to cusp. Inclination will be ~90° to ecliptic.

2.2.1 Summary of Mission Phases

Table 2 Mission Phase Summary

Parameter	Injection/ Separation Phase 0	Phase I	Phase II Apogee follows Magnetopause	Phase III Double Lunar Swingby	Phase IV
Orbit (Re)	1.2 x 12	1.2 x 12	1.2 x 30	8 x ~120	10 x 50
Orbit Period	24hr	~~~~24hr (synchronous)	3.6day	~30day	9.6day
Inclination	28.5°	10° (in the magnetic equator)	10°	~~N/A~~~0°	90° to ecliptic
Arg of Perigee	0° (daylight)	0°	Perigee ~6am to 2pm local time		0°
Max Eclipse	1hr	1hr	*	*	1.2hr
Eclipses per year	N/A	255	TBD	~~TBD~~0	4-6

Satellite separation	<10km	10km to 0.1Re	0.01Re to 1Re	1Re to 3-~~410~~Re	10km to 1Re
Phase duration	hours	0.6yr	0.3yr	0.2yr	0.8yr

.* The mission design and launch time constraints will limit the worst case eclipse duration is < 2hr.

2.2.2 Delta V Budget estimates

Table 3 DeltaV Summary

Maneuver	Phase	Nominal dV (m/s)
Inclination change	Separation	326
Spin plane boom deployment	Separation/I	44
Apogee Adjust	I	277
Apogee Adjust	II	110
Double Lunar Swingby/Phase IV initialization	III	90
Perigee maintenance	II	12
Launch window/Arg of Perigee adjust	Launch	35
Launch dispersion	Launch	12
Tetrahedron development	I, II, IV	120
Total		**~~982~~ 1026**
Capability		1100
Margin		~~1.8~~6.7%

2.3 Design Lifetime

[MMS47] Mission lifetime of two years.

Table 4

[MMS48] Selective redundancy

Table 5

[MMS49] Expendables sized for a dV of at least 1100m/s.

Table 6

2.4 Spacecraft Attitude Control.

A spin stabilized spacecraft is assumed with the following parameters:

2.4.1 Spin axis

[MMS2] Spin axis shall be offset from normal to ecliptic plane by 2-5°.

Table 7

Offsetting the spin axis by 2-5° prevents the body of the spacecraft from shadowing the electric field sensors.

2.4.2 Spin rate control

[MMS45] Spin rate controlled to 20rpm (±0.2rpm)

Table 8

2.4.3 Spin axis knowledge

[MMS46] Spin axis knowledge (post-processed) < 0.1°, spin phase knowledge (post-processed) < 0.1°. See also discussion of magnetic field instrument.

Table 9

2.4.4 Orbit determination:

[MMS3] Knowledge of individual spacecraft position is 100km except immediately prior to maneuvers when the requirement shall be TBDkm.

Table 10

2.4.5 Spacecraft stability

[MMS4] The spacecraft shall be stable in all mission phases.

Table 11

[MMS5] Prior to the deployment of the electric field booms the spin-to-tumble inertia ratio shall be > 1.04.

Table 12

The axial and spin plane booms will dominate the dynamics of the spacecraft after deployment. For analysis purposes the ~~axial~~ booms may be assumed to have the ~~following~~ properties defined in 3.4

2.4.6 Spacecraft Dynamics

[MMS6] The spacecraft manufacturer shall be responsible for calculating all dynamics interactions between the booms and spacecraft and for ensuring that all coning angle and nutation specifications are met within TBD times of eclipse exit & dV maneuvers.

Table 13

2.5 Electrical Power

Instrument orbit average power is given in the instrument summary Table 31 . It is anticipated that extended eclipses will be encountered during phased I, II, III & IV for the multiprobe spacecraft. A detailed analysis has been performed of eclipse duration throughout these phases. Eclipse duration will be limited to 120 minutes and instruments will be assumed to remain powered during eclipses. [MMS51] The spacecraft shall support eclipses with duration up to 120minutes with the instruments fully operational during eclipse.

Table 14

[MMS50] The spacecraft shall provide bus voltages within the range of 22V-35V (nominal 28V) to the instruments.

Table 15

2.6 Communications

2.6.1 Uplink/Downlink Frequency

[MMS7] The spacecraft shall use an X-band frequency for uplink and downlink.

Table 16

[MMS8] Command link shall to be maintained at maximum apogee.

Table 17

2.6.2 Groundstation stratagy

Commercial 11m groundstations will be used for phases I & II. DSN 34m HEF groundstations will be used for phases III & IV. [MMS9] In order to minimize groundstation cost, the spacecraft shall have the capability to store up to 14days of science data without loss and downlink the data at a rate of at least 1 Mbps but less than 2.2Mbps.

Table 18

The intent is to store data until the range to the groundstation is short enough to allow the data to be transmitted at a high rate.

2.6.3 Spacecraft EIRP

[MMS10] The spacecraft shall be capable of transmitting an EIRP (Effective Isotropic Radiated Power) at least 14dBW in the direction of the earth.

Table 19

[MMS11] The transmitter shall be capable of transmitting continuously for at least 4hours.

Table 20

2.6.4 Science data/Commanding Volume

The instrument complement will generate ~2 Gbits per day per spacecraft.
[MMS12] Commanding for the instruments shall be 100 bytes per instrument per day for each spacecraft.

Table 21

2.6.5 Tracking

The strategy for tracking the spacecraft and determining their orbit is currently under study. The options being considered are to use two-way Doppler or to require the spacecraft to fly high stability oscillators and use one-way Doppler[MMS44] . Two-way Doppler is the current baseline with a requirement of 100km for normal operations.

Table 22

2.6.6 Command and Telemetry Format

[MMS13] The uplink and downlink format shall be Consultative Committee for Space Data Systems (CCSDS) Advanced Orbiting Systems (AOS) format.

Table 23

2.7 Environments

2.7.1 Radiation Environment

[MMS43] The anticipated total radiation dose for all four phases for 2 years, without margin, for the MMS orbit is shown in Figure 1.

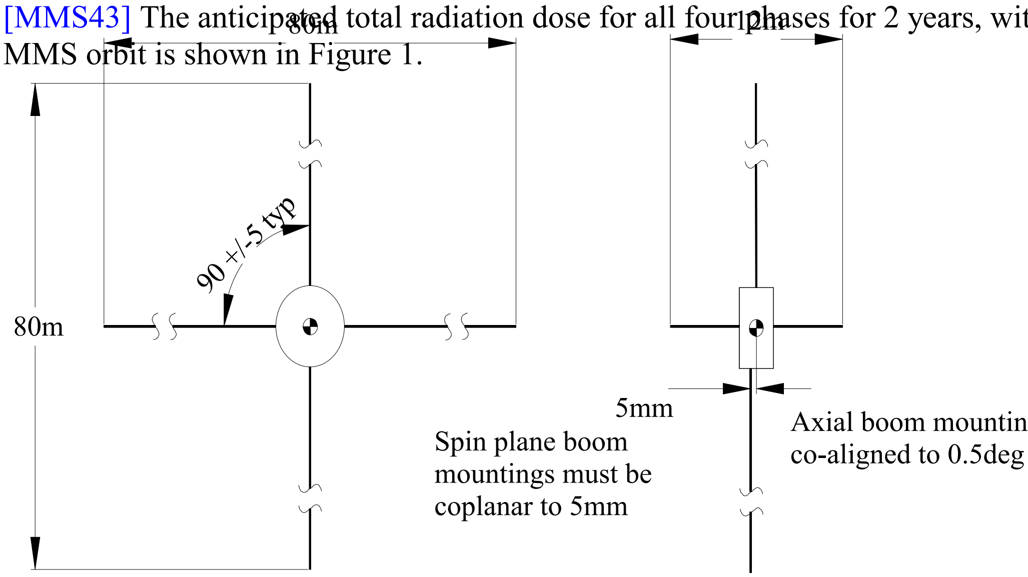

Figure 1 MMS Total Dose Curve

Table 24

2.7.2 EMC

Since the instruments measure very low levels of plasma energy the spacecraft must not disturb the surrounding plasma. [MMS14] The spacecraft exterior surface shall be an equi~~iso~~-potential surface with no point of the exterior surface more than 1Volt different than any other point on the spacecraft.

Table 25

This requirement applies to all surfaces including blankets and solar arrays.
The magnetic compatibility requirements are described in Magnetometer instrument section 3.5.1

2.7.3 Contamination

The Hot Plasma instrument and the Energetic Particle instrument are susceptible to particulate contamination and must be under high purity nitrogen purge ~~until launch~~. [MMS15] A break-away plumbing system shall provide purge to all spacecraft until lift-off.

Table 26

2.8 Mechanical

[MMS16] The spacecraft design shall be compatible with the launch environment for a Delta II 7925 and support the instrument allocations identified in Table 31.

Table 27

[MMS17] The mechanical design shall accommodate the fairing volume, center of gravity location, and stiffness requirements for the Delta II.

Table 28

[MMS18] The mechanical subsystem shall have provision to provide dedicated high purity nitrogen purge to the Hot Plasma and Energetic Particle instruments of all spacecraft through launch.

Table 29

[MMS19] The mechanical layout of the instruments shall provide unobstructed fields of view for the Hot Plasma and Energetic Particle instruments.

Table 30

3 Instruments

3.1 Instrument Summary

While the instruments have not been selected for the MMS mission, the science definition team has suggested a candidate list of instruments. For the purposes of this study the list of instruments will be as shown.

Table 31 Instrument Summary

	Magnetometer and Search coil	Hot plasma detector	Energetic Particles	Electric Field	Totals
Mass (kg) per unit	1.5	8.0	2.5	17.0	38.5kg
Quantity	1 each	2	1	1	
Operation Power (W) per unit	1.2	7.0	2.0	15.0	33.4W
Data Rate (Normal) per unit	5 kbps	6 kbps	2 kbps	5 kbps	24 kbps
Data Rate (Burst) per unit	10 kbps	32 kbps	2 kbps	65 kbps	104 kbps
FOV (deg)		10° x 360° (1)	10°x160°		
Size (cm)	18x10x8 (Elec.) 5x5 (Sensor)	20.3ϕx20.3	11x11x10	See description	

(1) For non-scanning version of the instrument. See description of instrument.

The Interspacecraft Ranging and Alarm System (IRAS) shall be considered a GFE'd instrument for the purposes of this document.

Table 32

See detailed description of the IRAS 3.6

Table 33 IRAS Summary

	IRAS
Mass (kg)	1.2kg electronics + 0.8kg antennas
Operation Power (W)	5 (avg), 15 (peak)
C&DH storage Rate	100bps
C&DH transfer rate	500bps
Size cm (LxWxH)	12 x 12 x 4

3.2 Hot Plasma Instrument

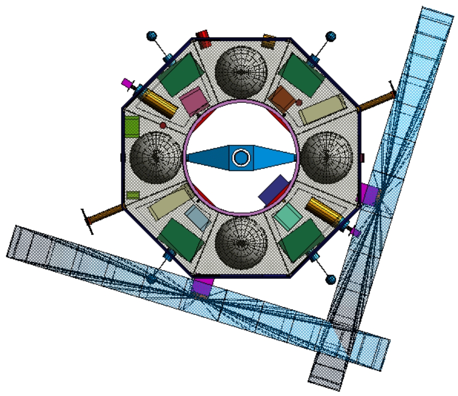

Figure 2 20rpm Design

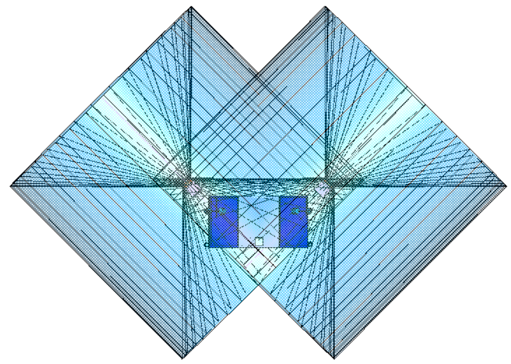

Figure 3 6rpm Design

The Hot Plasma instrument is used for measuring the number density and energy for particles in the range of 1eV to 30eV. The instrument will have a geometry factor of 0.01 cm^2-sr-eV/eV for ions and 0.005 cm^2-sr-eV/eV for electrons, an energy resolution of 4 (E/ΔE) and an absolute accuracy of 5-10% with a relative accuracy of 1% between spacecraft. The instrument can be built in one of two configurations: a non-scanning instrument with an field of view of 10° x 360° (shown in figure 3) and a scanning version with an instantaneous field of view of 10° x 360° but the beam can scan ±45° (shown in figure 3). The two of the non-scanning instruments can provide 4π steradian coverage at 0.75seconds time resolution if they are mounted as shown in figure 3. The time resolution is inversely proportional to the spin rate. For slower spinning spacecraft, an electrostatically scanning instrument has been developed which can provide the time resolution independent of spin rate. To provide 4π steradian coverage, two instruments are mounted at 180° to each other as shown in figure 2. In each case the instrument sensor body is approximately a cylinder 8inch diameter and 8inch long. The non-scanning instrument has been baselined for the MMS mission. In the event that the spacecraft is not capable of meeting the pointing requirements at 20rpm it is possible to use the non-scanning instrument. This instrument will require a constant high purity nitrogen purge from integration to launch. A purge port will be provided at the instrument interface.
It can be assumed that the instrument is capable of a Mil-Std-1553 or other high level interface.

3.3 Energetic Particle Instrument

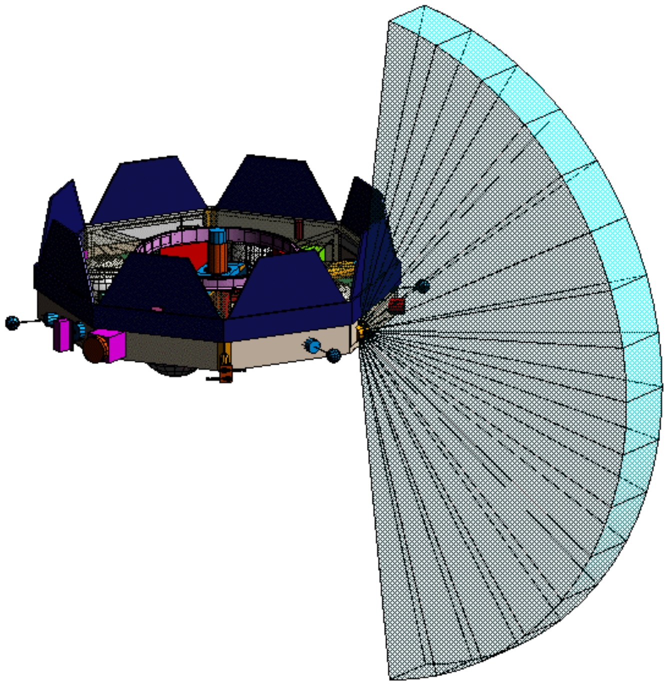

Figure 4 Energetic Particle Field of View

The energetic particle instrument measures number density, angle, energy and mass/charge for particles in the range 30keV-300keV. The instrument has a field of view of 10° x 160° as shown in the following figure. The instrument uses the spin of the spacecraft to give almost complete coverage. There is, however, a 20° hole in the coverage around the spin axis. This instrument will require a constant high purity nitrogen purge from integration to launch. A purge port will be provided at the instrument interface. It may be assumed that the Energetic Particle Instrument has a Mil-Std-1553 or other high level interface.

3.4 Electric Field Instrument

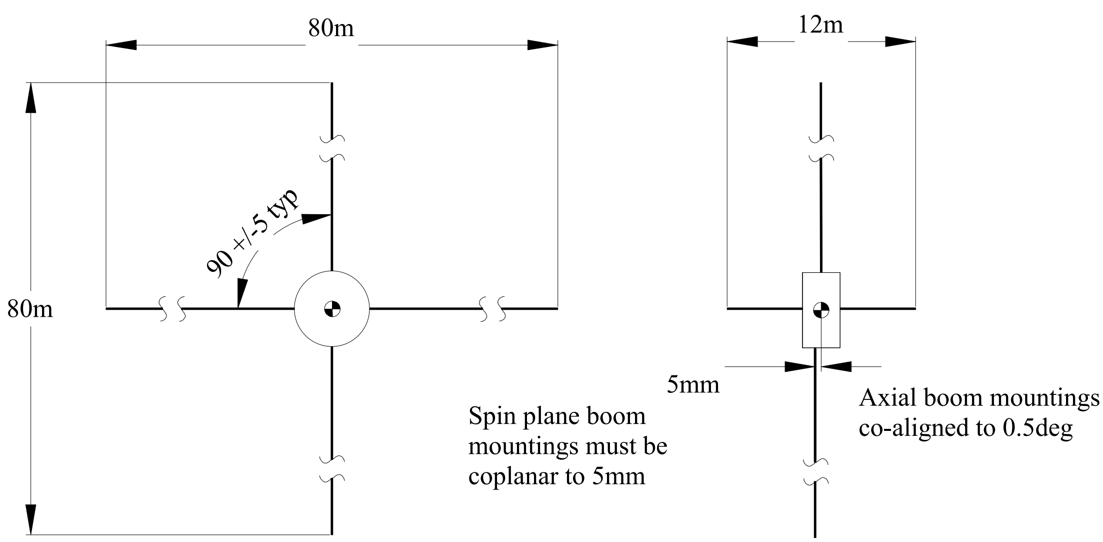

Figure 5

The electric field instrument consists of spin plane booms, axial booms and an associated electronics box.

3.4.1 The axial booms

- Each axial boom has a 6m deployed length and less than .35 m stowed length along the deploying axis (not including the sensor sphere). Stowed volume less than 0.0075 m^3.
- Capable of supporting a single 0.15 Kg mass (10 cm diameter sphere) at the tip end and a cable bundle along the entire length. The minimum (fixed base) first mode bending and torsional frequency is 1 Hz.
- Deployed boom tip precisely positioned and stable within a full cone angle of 0.2° with respect to a known reference plane while under the influence of thermal effects consistent with a low earth orbit.
- Boom and deploying mechanism must be less than 2.5kg.
- [MMS42] Axial booms may be packaged end-to-end or side-by-side. In either configuration, the mounting scheme must allow for the center of gravity of the deployed boom to be on the geometric spin axis.

Table 34

-
- [MMS21] Mounting surfaces for the two axial booms shall be co-aligned to within 0.1°

Table 35

3.4.2 The spin plane booms

- Deployed wire length greater than 40 m, stowed length less than 0.35 m (not including the sensor spheres).
- Capable of supporting two 10 cm diameter sensor spheres of 0.15 Kg each along the wire length, one at 75% radially out and the other at the tip.
- Wire bundle for power and signal equivalent to 0.03 kg/m linear density
- Boom and deploying mechanism less than 2 kg
- The attachment point at the spacecraft for the booms must be at 90° ±5° around the circumference of the spacecraft.
- ~~The attachment points must be in a plane and that plane must contain the spacecraft center of gravity. The center of gravity may be up to 0.25inches out of the plane~~.
- The mounting surfaces for the booms to be co-planar to 0.1inch. [MMS22] The spacecraft CG shall be remain within 4 inches axially of the boom spin plane.

Table 36

3.4.3 Electronics Box

The electric field electronics box is estimated to weigh 2.5kg and be approximately 6”x6”x5”. It may be assumed that the Electric Field instrument has a Mil-Std-1553 or other high level interface.

3.5 Magnetic Field Instrument

The magnetometer instrument consists of a sensor head and an electronics unit.

3.5.1 Spacecraft magnetic field

[MMS41] The spacecraft must not generate magnetic fields of more than 0.1nT DC and 0.03nT AC (>1Hz) at the sensor head.

Table 37

~~In light of these requirements I~~It is up to the spacecraft manufacturer to determine the length of the ~~whether he uses a~~ boom to mount the magnetometer head away from the main body of the spacecraft ~~or if~~and the magnetic cleanliness program~~of the spacecraft is such that a boom is not necessary~~. ~~In any event, T~~The mass and cost of a boom will be borne by the spacecraft.

3.5.2 Magnetometer pointing accuracy

The spacecraft pointing requirements are based on the requirements of the magnetometer. [MMS39] The fundamental MMS pointing requirement is to be able to know the orientation of the magnetometer sensor relative to the spacecraft body co-ordinate system to ±0.25°.

Table 38

3.5.3 Co-alignment of Magnetometer and Electric Field booms

[MMS40] The alignment of the Magnetometer sensor head and the electric field booms must be known to 0.5° in the on orbit configuration.

Table 39

3.5.4 Correlation of Magnetometer data and Electric Field data

[MMS23] The spacecraft timing system shall have sufficient resolution to allow correlation of electric field telemetry and magnetic field telemetry to less than ~~400~~50μs.

Table 40

3.5.5 Correlation of Magnetometers between spacecraft

[MMS24] The spacecraft C&DH subsystem shall allow correlation between spacecraft of magnetometer data.

Table 41

[MMS25] The IRAS shall allow time transfer to ~~400~~50μs between spacecraft.

Table 42

3.6 Interspacraft Ranging and Alarm System

The IRAS is an RF based ranging system that works on similar priciples to the GPS. Each spacecraft will transmit a message in turn, and each of the other spacecraft will time-tag the receipt of the message. From the differences in the times of the received and transmitted messages an accurate measurement of the spacecraft separations can be made.

3.6.1 IRAS requirements

1. [MMS26] The IRAS shall measure the distance between the five Magnetospheric Multi-Scale spacecraft.

Table 43

2. [MMS38] The IRAS shall measure the distance between the spacecraft with accuracy better than 1%.

Table 44

3. [MMS27] A low speed serial message shall be passed from the IRAS to the spacecraft C&DH processor.

Table 45

4. [MMS37] The message shall contain the following information: distance to other members of the formation, alarm status of each spacecraft, thruster firing status of each spacecraft, internal IRAS health and safety.

Table 46

5. [MMS28] Maximum time from alarm message input to transmitting IRAS to time alarm signal output from receiving IRAS shall be 3 sec.

Table 47

6. [MMS29] The IRAS system shall be operational at all times in all mission modes.

Table 48

7. [MMS30] The IRAS system shall be capable of correlating time among the five spacecraft to less than 50µs

Table 49

3.6.2 IRAS to C&DH interfaces

3.6.2.1 Telemetry and Alarms

[MMS31] The IRAS shall have a Mil-Std-1553 interface to allow low speed telemetry and high speed alarms to pass among the spacecraft.

Table 50

Table 51

Alarm	C&DH response	Comments
~~Interesting s~~Science Event	Send message to all instruments within 0.5second of receipt	Other spacecraft can go into a high speed data capture mode on receipt of this command
One spacecraft has aborted a thruster firing	Receiving spacecraft aborts thrusting	During maneuvers the five spacecraft must remain together. In the event that one of the spacecraft has to abort a maneuver, this message allows other spacecraft to abort also.

[MMS32] The Mil-Std-1553 schedule table shall allow for '~~interesting s~~Science Event' messages to be passed from any of the instruments to the IRAS within 0.5s.

Table 52

[MMS33] The spacecraft C&DH shall pass a message to the IRAS in the event that that spacecraft has aborted a maneuver.

Table 53

3.6.2.2 Timing

[MMS34] The C&DH shall pass to the IRAS a discrete RS-422 timing pulse that is correlated with the fundamental timing system of the spacecraft.

Table 54

[MMS35] The pulse repetition frequency shall be 1Hz.

Table 55

The IRAS will use the pulses to correlate time among the spacecraft. [MMS36] The correlated time information shall be telemetered from the IRAS as Mil-Std-1553 messages.

Table 56

There is no requirement for the spacecraft to read or use the correlated time messages. The correlated time messages will be used on the ground to correlate time between the spacecraft instruments.

3.6.3 IRAS antennas

The IRAS will have two S-band antennas that are mounted to cover 4π steradian. The locations of the antennas will be negotiated between the spacecraft manufacturer and the IRAS designers.